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MULTIDISCIPLINARY ANALYSIS FOR A CIVIL AIRCRAFT CONCEPTUAL SIZING

Ohad Gur , Gali Lazar , Aharon Yaniv
IAI - Israel Aerospace Industries, Ben-Gurion Airport, Israel
Abhishek Burman
NAL - National Aerospace Laboratories, Bangalore, India

ABSTRACT

As part of conceptual design effort for a civil air-vehicle many sizing decisions should be taken. These decisions should be taken despite existence of chicken-or-the-egg dilemmas, i.e. deciding the aircraft configuration sizing parameters without knowing its appearance and performance, and deciding its appearance and performance without knowing its sizing parameters. The paper gives a review of a multi-disciplinary framework, which uses modeFrontier® environment. This framework consists of several software modules which were integrated into a single synergic tool. In the current scope this design tool is used to decide the aircraft wing loading while taking into account various performance constraints. The importance of the multidisciplinary analysis tool is demonstrated and its ability to handle large number of cross-influences is discussed. The existence of several disciplines integrated into a single multidisciplinary tool may introduce some difficulties, mainly if engineering intuition is involved.

Key Words: Multidisciplinary, Conceptual, Design

NOMENCLATURE

AR	Aspect ratio
a	Speed of sound
b	Span
C_{app}	Approach speed factor
C_D	Drag coefficient
C_{D0}	Zero-lift drag coefficient
C_{Df}	Friction/Form drag coefficient
C_{Di}	Induced drag coefficient
C_f	Friction coefficient
C_L	Lift coefficient
C_{Lmax}	Maximum lift coefficient
D	Drag
D_0, D_i	Zero-lift and induced drag, respectively
e	Oswald coefficient
FF	Form factor
H	Altitude
IF	Interference factor
L	Lift
M	Mach number
q	Dynamic pressure
R	Range
S_{ref}	Reference area
S_{wet}	Wetted area
T	Thrust
TF	Thrust factor
Th	Throttle
$TSFC$	Thrust specific fuel consumption
V	Velocity
V_{app}	Approach speed
V_{st}	Stall speed
W	Weight
W_i, W_f	Initial and final weight, respectively
ρ	Air density

1. INTRODUCTION

Designing civil air-vehicle is an immense task which requires a multidisciplinary perspective. This is especially true during the conceptual design phase. At initial design phase the designer owns most of the design freedom and its influence over the final design is probably the greatest. These two important issues (design freedom and design authority) give extra importance to the conceptual design phase.

Figure 1 presents these issues and shows one of the design paradigms. While owning such an authority and freedom, at later stages, the designers are confined to a predefined design without ability to alter it. By involving more disciplines at earlier stages of development, the knowledge about the final design is increased while the design freedom stays high for later design stages.[1] The solution is available through multidisciplinary analysis and optimization (MAO) at the initial design stages, especially at the conceptual design phase.

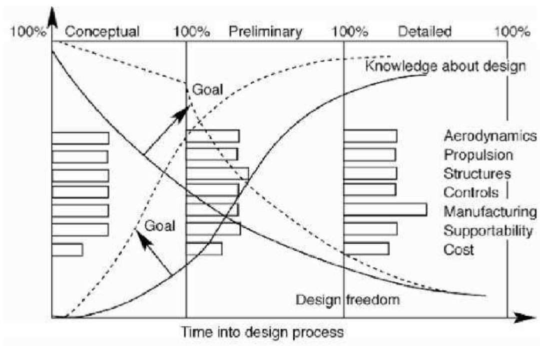

Figure 1 Design freedom and design knowledge through R&D process [1]

As part of a feasibility study for a new civilian aircraft, a sensitivity study which uses multidisciplinary analysis (MA) was conducted to enable a better decision making regarding the aircraft conceptual sizing. The following paper describes this MA effort.

2. THE CONCEPTUAL SIZING DILEMMA

Trying to establish an air-vehicle size is somewhat equivalent to the chicken-or-the-egg dilemma. Without really knowing the aircraft shape, its size and resulted performance should be evaluated. On the other hand, without really knowing its size and performance, the aircraft shape should be decided. This dilemma is represented as a cartoon by Bob Sandusky, Northrop (Figure 2) taken from Ref. 2 .

Figure 2 Conceptual Sizing Dilemma Cartoon [2]

One of the major decisions is the aircraft weight over reference area, i.e. wing loading, W/S_{ref}. This fundamental sizing parameter influences the aircraft performance through all flight regimes. The aircraft range, *R,* can be evaluated using the well-known Breguet equation:

$$R = \frac{V}{TSFC}\frac{C_L}{C_D}\ln\frac{W_i}{W_f} \tag{1}$$

where *V* is the airspeed, *TSFC* is the engine thrust specific fuel consumption (fuel weight flow over thrust force), *CL* and *CD* are the lift and drag coefficient, respectively, and *Wi* and *Wf* are the initial and final aircraft weight, respectively.

A simple drag polar is assumed. It consists of zero-lift drag coefficient, *CD0,* and induced drag coefficient, *CDi* :

$$C_D = C_{D0} + C_{Di} \tag{2}$$

The induced drag coefficient depends on the square of the lift coefficient, C_L, Aspect ratio, AR, and Oswald coefficient, *e:*

$$C_{Di} = \frac{C_L^{\ 2}}{\pi\, AR\, e} \tag{3}$$

Using this conventional formulation it is easy to show that the maximum range is proportional to the following range parameter:[3]

$$R_{opt} \propto \frac{1}{W}\left[\frac{4\pi\, e\, AR\left(W/S_{ref}\right)^2}{\rho^2 C_{D0}^{\ 3}}\right]^{\frac{1}{4}} \tag{4}$$

where *W* is a typical weight of the aircraft during cruise and *Sref* is the reference area, typically the wing area is used.

Equation (4) implies that higher wing loading, W/S_{ref}, contributes to an increased range.

On the other hand, lower wing loading contributes to shorter take-off and landing distances. To demonstrate this, a simplified analysis for the approach speed, *Vapp,* is given.

The approach speed is considered to be higher than the stall speed, *Vst,* by certain margin, typically 20% -30%, thus:

Note that maximum lift coefficient, C_{Lmax}, is addressed in this simplified analysis, as a constant parameter. The decision of the maximum lift coefficient and designing high lift devices which enable to produce such a coefficient is a complicated process and should be treated carefully.

Generally low approach speed is desirable due to decreased landing distance and lower acoustic signature of the aircraft at landing. As stated above, this means that low wing loading is desirable.

This simple analysis of range versus approach speed displays one of the compromises required from a designer at the conceptual design phase.

The current effort focuses on assisting the decision makers and designers to choose an appropriate wing loading while taking into consideration the various design requirements.

3. THE DESIGN FRAMEWORK

The design framework was built using ESTECO modeFrontier® framework. modeFrontier® is a multi-disciplinary and multi-objective optimization and design environment which enables an easy integration of existing analysis modules into a modified design tool.

To explore the cross influences of the design, various analyses were integrated into modeFrontier®. The analyses had been self-developed by IAI, and were not developed as part of the current effort. Thus, any other preliminary design and performance estimation tasks use the same analysis modules, but not through the current modeFrontier® integrated form. Figure 3 shows the design environment screenshot containing integration of the various analyses.

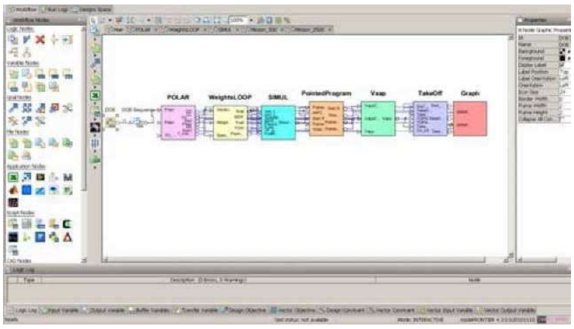

Figure 3 modeFrontier® design framework

The following analyses were integrated:

a. Drag polar estimation procedure. This is used to estimate the aircraft drag as function of its geometry. Both cruise configuration and take-off configuration polars are produced by this module.
b. Weight estimation procedure. The weight estimation relates the aircraft geometry, flight characteristics and max take-off weight, to its basic empty weight (BEW). It is based on extensive statistical and empirical models for every aircraft element (wing, landing gear, systems, etc.).
c. Point performance analysis which is used for calculating the rate-of-climb (ROC) at initial cruise condition.
d. Point performance analysis which is used for approach speed calculation at landing.
e. Point performance analysis which is used for specific range calculation at mid cruise conditions (50% fuel).
f. Mission analysis simulation procedure used to calculate the range at specified fuel weight.
g. Mission analysis simulation procedure used to calculate the fuel needed for a given range.
h. Take-off simulation used for the FAR25 take-off balanced field length (BFL) and second segment climb gradient calculation.

The two main analyses are the aerodynamic and weight estimation. These will be discussed in a two separate sub sections. In addition, an engine model is available. Note that the current design framework uses a predefined engine model. This will also be discussed later.

Integration of the various analyses into a unified tool enables a multidisciplinary analysis, while taking into consideration the cross influences of each configuration parameter on the others. For example, the weight estimation procedure uses the wing area, span, and sweep angle as part of many geometric parameters which determine the aircraft BEW (basic empty weight). The same parameters are also used as an input to the drag polar estimation, thus influencing the mission and point-performance analyses. Many other cross influences exist such as wing area influence on empennage sizing, and consequently on its weight and drag.

3.1 Aerodynamic model

The aerodynamic model is used to estimate the aircraft drag polar, i.e. drag coefficient, C_D, vs. lift coefficient, *CL*. The model is based on semi empirical methods and contains 4 main contributions:

a. induced drag
b. friction/pressure drag
c. profile, lift dependent drag
d. wave (transonic) drag

The induced drag coefficient, CDi, estimation is based on Eq. (3). Where Oswald coefficient, e, is estimated based on IAI engineering experience. Some examples for typical values for *e* can be found in available literature.[5] According to Ref. 5, the Oswald coefficient for "high-subsonic jet aircraft" is *e* = 0.75-0.85.

Friction drag is calculated according to the wetted area, S_{wet}, and friction coefficient, *C*, of the various aircraft components. The friction coefficient is calculated according to the component average Reynolds number.

$$C_{Df} = \frac{S_{wet} C_f}{S_{ref}} FF\ IF \tag{7}$$

Two factors are applied to the friction drag; form factor, *FF,* and interference factor, *IF*. The form factor represents the contribution of the pressure drag and it is based on the thickness ratio, *t/c,* for lift carrying surfaces, and on the fineness ratio, *l/d,* for fuselage and nacelles. A thorough discussion regarding available form factors models is available in Ref. 6. The interference factor, *IF,* reflects the interference drag of the installed component. The interference factor calculation is based mainly on DATCOM method.[7]

Profile lift dependent drag is estimated by 2-D CFD analysis of the wing airfoil. Using the 2-dimensional lift influence on the 2-dimensional drag rise, the 3-dimensional drag coefficient is corrected. Although this is a simplified model it captures the lift dependent profile drag.

The wave drag component is estimated using a semi-analytic model. [5] This model takes into account the lifting surfaces thickness ratio, airfoil technology, supercritical technology, and the wing sweep angle.

The 4 drag components (induced, friction/pressure, profile, and wave) are calculated using Excel spread sheet and visual basic macro code. Than a formatted aerodynamic ASCII file is produced as an input to the other analysis models. The excel spread sheet enables the use of various geometric parameters as input and gives great flexibility and robustness to the resulted aerodynamic data.

3.2 Weight Estimation Model

The weight estimation model calculates the aircraft basic empty weight (BEW) which together with the payload and fuel weight, add up to the takeoff gross weight (TOGW).

The full configuration BEW estimation is done by adding separate contributions of the various aircraft components. Partial list of these components is: main wing, horizontal tail, vertical tail, fuselage, nacelles, landing gears, dry engine, fuel system, propulsion system, flight control system, hydraulic and pneumatic system, electric system and wiring, instrumentation and avionics system, aircraft pressurization and deicing system, environment control system, oxygen system, auxiliary power unit (APU), furnishing, operational items, and paint.

For each component 2-5 different models exist. Here comes in hand IAI experience and best engineering judgment, which take into account an educated average of the various model results.

The weight estimation model is based on a collection of various analytic, semi-analytic, and empirical models.

Several different sources of knowledge are used: design papers such as Beltramo et-al[7] and Svoboda[8], classic design test books such as Roskam[9] and Torenbeek[5], estimation software such the USAF DATCOM[10] and NASA GASP[11], and on line tools such as Ilan Kroo's Aircraft Design website[12]. Note that some of the mentioned resources cite each other. One important source of information is the accumulated experience with other configurations which were designed and constructed by IAI, thus the weight estimation model contains a large legacy of know-how.

An important feature of the weight module is a built-in convergence loop which finds the basic empty weight (BEW) of the configuration. The weight model uses the take-off gross weight (TOGW) as an input parameter. e.g the wing structural weight depends on the aircraft TOGW. This means that the weight estimation model calculates the aircraft weight while depending on a pre-estimated weight. This contradiction is easily solved by a simple iterative loop which uses the calculated BEW to update the aircraft TOGW and calculate again an updated BEW until convergence.

3.3 Engine Model

Some aircraft conceptual designs start from a given engine which was chosen as a basis for the new design. Again, this implies a dilemma of choosing an engine for an unknown sizing of the configuration. Nevertheless, some design features must be decided - engine type and size are two of these design anchors. In some cases, engine choice is being altered during the conceptual phase, but altering engines may cause major influence on the design.

The predefined engine characteristics are typically received from the engine manufacturer. The engine performance model is based on a given engine data base, which contains thrust, *T,* and specific fuel consumption, *TSFC,* curves as function of the engine throttle setting, *Th,* cruise altitude, *H,* and Mach number, *M.*

In the conceptual design phase the designer keeps some design freedom of changing the engine characteristics, in order to examine the influence of his initial choice. This freedom is exercised in the current model using a thrust factor, *TF.* This factor is used to flex the basic engine into a 'rubber' engine. Increased factor implies an increased thrust, increased engine dimensions (which influence the aircraft aerodynamics), and increased engine weight.

In the current effort thrust-to-weight ratio, *T/W,* is used as a sizing parameter. It defines the ratio between maximum static thrust at sea level, standard conditions, to the aircraft maximum take-off weight. A typical value for a civil aircraft is $T/W = 0.3 - 0.4$.

It seems that an increased thrust factor, *TF,* will increase the thrust-to-weight ratio. Still, the thrust factor also increases the aircraft weight (enlarged engine), thus the thrust to weight ratio can vary differently or even remain the same. Such a counter effect can be checked only in a multidisciplinary design framework. Note that while modifying the thrust, *T,* or thrust-to-weight ratio, *T/W,* the baseline engine specific fuel consumption, *TSFC,* is assumed to remain unchanged.

3.4 Performance Analysis

The design framework comprises several performance analyses. These analyses include a mission analysis, Simul, ground performance analysis, and point performance analysis.

Simul is a legacy code written in FORTRAN. It has been widely used as a mission analysis tool by IAI. The user interface of Simul is ASCII input and output files, thus it is easily integrated into the modeFrontier® framework.

The aircraft aerodynamic model contains a specialized interface module, written using Visual Basic, which produces a drag polar input file automatically according to Simul required format. In this way changing a geometric parameter (e.g. wing span) updates the input file for the mission analysis.

Similar interface exists for the take-off and landing field performance analysis, thus the aerodynamic model produces automatically input drag polars for the field performance model. This model is written and developed by IAI using Visual Basic. Some code updates were needed for the implementation in modeFrontier® due to the use of graphic user interface (GUI) in the original code.

The point performance analyses are done partially by specialized FORTRAN software (using the same input files as Simul mission performance code) and partially by Excel spread sheets (e.g. the approach speed estimation). In both cases the integration into within modeFrontier® is straightforward.

4. SENSITIVITY STUDY

A sensitivity study was conducted to examine the influence of two main sizing parameters: aircraft wing loading, *W/Sref,* and thrust to weight ratio, *T/W,* on multiple performance characteristics. The study produces two dimensional charts with the two sizing parameters as the horizontal and vertical axis and curves which represent constant performance characteristic.

These charts were produced by running the integrated design framework in an analysis mode and the results were post-processed in Matlab. The authors are aware of the fact that such a design framework can be easily used for optimization (see Ref. 13) but for the current scope, the design framework is used just as an analysis tool.

A nominal configuration was defined with a typical wing loading of W/S_{ref} = 100 psf (440 kgf per sq.m) and a take-off thrust/weight ratio of *T/W* = 0.32 . To methodologically investigate the design sensitivity to these parameters, two separate studies were conducted. The first consider the aircraft span to be constant with the wing area changing, thus varying the wing aspect ratio without altering the wing span. A second study was conducted with a constant aspect ratio, thus the wing span is changed along with the wing area. This can be referred as homothetic (scaled) wing geometry transformation.

These studies (constant span and homothetic transformation) do not include the influence of the thrust-to-weight ratio on the aircraft BEW, thus the engine weight is not influenced by updating the thrust-to-weight ratio. Although this is somewhat inaccurate, the results look much more intuitive and easy to understand by the decision makers. In the last sub-section of the current section a study containing the BEW / thrust-to-weight ratio cross-influence is presented and discussed in details.

The entire study was done using the nominal flight conditions of 35,000 ft cruise altitude, and takeoff from S.L. at ISA+15 deg.

4.1 Constant span

As mentioned above the first study uses constant span. While the wing loading parameter is changed the wing aspect ratio varies as well.

Two important constraints are the aircraft rate-of- climb at initial cruise altitude and weight, and the one engine inoperative (OEI) second segment climb gradient at take-off. The first indicates the aircraft ability to accomplish its mission at the required cruise altitude and the second is a safety requirement to ensure sufficient rate-of-climb right after the take-off. Two sets of lines representing these two parameters are presented in Figure 4 on a thrust-to-weight versus wing loading chart. The nominal design is designated as a red diamond.

The initial cruise altitude and weight minimum recommended rate-of-climb is 250 ft/min. Obviously for the nominal design, where it is more than 1,000 ft/min this does not impose a constraint on the design. According to FAR 25 the second segment climb gradient should be more than 2.4%. According to Figure 4 it is higher than 7% thus both initial cruise ROC and 2nd segment climb gradient do not constrain the nominal design and the wing loading can be easily increased without exceeding these two design constraints.

Note that this conclusion is valid also for the following case studies. This is the reason that the initial cruise rate-of-climb and 2nd segment climb gradient curves are not presented in the following cases.

An important parameter is the range parameter *ML/D* which indicates the aerodynamic part in

Breguet range equation. Equation (1) can be rewritten in the following format:

$$R = \frac{a}{TSFC} M \frac{L}{D} \ln \frac{W_i}{W_f} \quad (8)$$

where a is the speed of sound, M is the flight Mach number, and L/D is the lift to drag ratio.

Figure 5 shows the carpet plot for the aerodynamic range parameter, constant span case. Note that the chart was produced for a 50% fuel capacity (half way cruise leg). It is clear that as the wing loading increases the range parameter (thus the range itself) increases as well. This was shown above in Eq. (4).

Further confirmation of this known result is presented in Figure 6. This chart includes three separate range figures-of-merit (all for the constant span case). The first is the previous range parameter *ML/D*. Second is the instrument flight rules (IFR) range in a standard mission with maximum fuel (including all appropriate fuel reserves) and nominal passengers load configuration. The third set of lines represents the required fuel weight to accomplish a standard 500 NM mission, again with all required reserves and nominal payload.

While the range parameter *ML/D* is represented by vertical lines, the range and fuel weight, which represent the aircraft range in a more comprehendible way, are represented by curved lines due to the cross influence of the thrust-to-weight ratio. Still, all 3 parameters exhibit a tendency to increase the aircraft range as the wing loading is increased.

The comparison of a relatively simple parameter such as *ML/D* range parameter to more comprehensive mission results introduces both the strength and troublesome nature of MA framework. While the decision makers are used to think in a simplified way by analyzing each discipline at a time, the MA framework enables much more elaborate parameters to be displayed. This gives the designer large number of parameters to consider but at the same time adds complexity to his decisions.

Both Figure 4 and Figure 6 suggest that the nominal design should be altered, thus the wing loading will be increased. This will enhance the range while enabling safe climb after take-off and positive rate of climb at initial cruise conditions. As shown above an increased wing loading is constrained by the ground performance of the aircraft.

Figure 7 shows the field performance results and the influence of *W/Sref* and *T/W* variation on the nominal design. This chart includes 3 types of lines. Range parameter *(M-L/D)*, BFL (balanced field length), and approach speed (V_{app}) at maximum landing weight, which is considered to be 90% of the take-off gross weight (TOGW).

Figure 7 exhibits the main contradicting trends for the current configuration; the field performance and range. As the wing area is decreased the wing loading rises; both approach speed and BFL increase. At the same time, as the wing loading rises the aircraft range is improved. Thus, this chart is used by the designer to inspect the influence of altering the nominal design.

4.2 Constant aspect ratio

As stated above, an additional study was conducted using constant aspect ratio assumption (homothetic wing transformation).

Figure 8 is equivalent to the constant span Figure 6 for the 3 range parameters: *M L/D,* range for maximum fuel configuration, and fuel needed for 500NM mission. For the homothetic wing scaling study the nominal design wing loading is in the proximity of an extremum, concerning the range and mission fuel weight. By an increased wing loading the range is not improved. This contradicts the trend of Eq. (4), but again by using MA framework which correctly represents the physical world, results that are not straightforward may be revealed.

$$D_i = \frac{(W/b)^2}{\pi \ eq} \quad (9)$$

Where b is the wing span and q is the dynamic pressure.

On the other hand the zero-lift dimensional drag, Do, is:

$$D_0 = qS_{ref}C_{D0} \quad (10)$$

Equations (9) and (10) show contradicting trends. For lower span (and lower wing area) the induced drag is increased while for lower wing area the zero- lift drag slightly decreases. Note that except for the wing, all other aircraft components zero-lift drag is not influenced by the change of the reference/wing area.

This means that the nominal design is in a region where a reduction of the span reduces the range, although the zero-lift drag is decreased.

For the case of constant span with the same aspect ratio, as the wing loading increases, the wing area decreases while the wing span remains the same.

This implies that the range will increase - the induced drag remains the same and the zero-lift drag is decreased.

This simplified discussion does not reveal the entire picture. In addition to the aerodynamic influence of the decreased span and increased wing area, one should remember that cross influences with the weight model also exist. This means that although the wing span decreases, thus the induced drag rise, the aircraft BEW decreases as well. Consequently it is possible that for a decreased span the induced drag increase will not reduce the range if the weight-to- span ratio is kept.

Figure **9** exhibits the same data as in Figure 7 but for the constant aspect-ratio (homothetic wing transformation) case. It shows both the aerodynamic range parameter, *M L/D,* and the field performance BFL and approach speed parameters.

As discussed above the range parameters exhibit almost neutral sensitivity to the wing loading, thus increasing the wing loading causes small change of range and above a certain value, even deterioration. The field performance (BFL and *Vapp)* tend to behave similar to the constant span case. As the wing loading increases both BFL and approach speed are increased, thus the field performance deteriorates. This suggests that for the homothetic wing case, an increase in wing loading is not recommended.

4.2 Thrust-factor weight cross-influence

As stated above the thrust factor (or the thrust-to- weight ratio) influence on the weight model was not included in the study presented so far. This means that for higher thrust factor, which implies higher thrust and engine weight, the aircraft BEW did not increase. In the current framework it is relatively simple to add this kind of interaction but the results might be somewhat confusing.

Figure 10 shows the same curves as in Figure 6: the range parameter, *M L/D,* range for maximum fuel configuration, and fuel needed for 500NM mission, all for a constant span case. Figure 10 differs by introducing the cross-interaction of the thrust factor and the weight model, which influences mainly on the *ML/D* parameters. While in Figure 6 these curves are vertical, by introducing the correct cross- interaction the curves tilt. This inclination is confusing due to the nature of the *M L/D* parameter. Apparently the aerodynamic range parameter does not contain information concerning the aircraft thrust. Still, if the aircraft BEW increases due to higher engine weight (increased thrust-to-weight ratio), higher wing loading (lower wing area) is required to maintain the same range.

Intuitively it seems as a sufficient explanation for the curves inclination. Still, the fact that the aircraft weight, *W,* plays an important role (and in reverse influence) for both: wing loading, *W/Sref,* and thrust- to-weight ratio, *T/W,* might cause difficulties.

Due to the nature of the results, it was decided that this cross interaction will not be activated. Although it causes some inaccuracies, the former output charts (Figure 6 and Figure 7) are easier to understand. If the current design framework will be used to optimize certain parameters (e.g. wing loading) it is desirable that all possible cross influences will be activated, thus the models will represent the physics in most accurate manner. If the design framework is used to produce general sizing charts such as the current carpet charts, it might be clever to loose some of the model accuracy and to produce more comprehendible results. Again, as always, it is the role of the designer to decide.

5. CONCLUSIONS

A multidisciplinary design framework was integrated to assist in the conceptual design effort. The current paper focuses mainly on the wing loading and thrust-to-weight ratio decision at the very early stages of a civil aircraft design.

The nominal thrust-to-weight ratio is determined according to the initial engine choice, thus altering this parameter may cause engine replacement. This is a major design change, thus the initial engine choice is crucial. It is clear that engine choice is much more restrictive than wing sizing, thus the thrust-to-weight ratio is more rigid compared to the wing loading.

Design framework is based on modeFrontier® environment and contains several software tools which were developed as stand alone tools. These modules consist of FORTRAN legacy codes along with Visual-Basic, Excel spread sheets, and Matlab components.

Results of the current effort are presented using thrust-to-weight versus wing loading carpets. On these plots, several curves representing different parameters and constraints are drawn: mission and range parameters, field performance parameters, and point performance constraints.

The main contradicting aspects for the aircraft wing loading sizing are the range versus the field performance. As the wing loading is increased, the field performance deteriorates, i.e. The balanced-field-length increases and the approach speed increases. At the same time as the wing loading increases, the range parameters are improved. i.e. The aerodynamic range parameter *(ML/D)* and the mission range increase, while the fuel needed for the 500 NM mission is decreased.

Using the trend study and the resulting charts, decision can be taken regarding modification of a nominal design.

It is shown that homothetic wing change is not beneficial considering the aircraft performance. Constant span while altering the aircraft aspect ratio has more potential concerning the aircraft drag reduction and range increase.

Taking into account cross-influences and studying simultaneously the influence of geometry parameters on aerodynamics, weight, and performance is possible just if an integrated multidisciplinary design framework exists.

Still, this benefit can cause an overload of information and in some cases the results are hard to comprehend. An example for such cross-influence between the thrust-to-weight ratio and engine weight is demonstrated and discussed. For a decision making purpose this cross-influence should be neglected. It is obvious that if an optimization is conducted, all possible cross-influences should be introduced, thus the physical credibility of the framework is increased.

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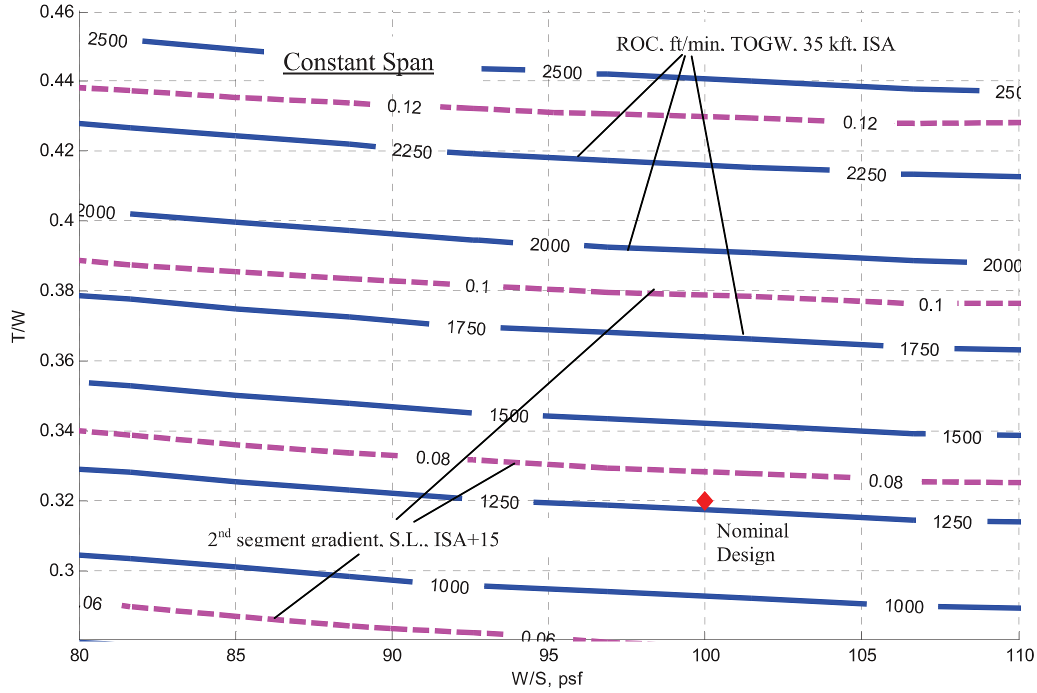

Figure 4. Constant span wing loading and thrust/weight ratio carpet. 2nd segment climb gradient and initial cruise ROC curves.

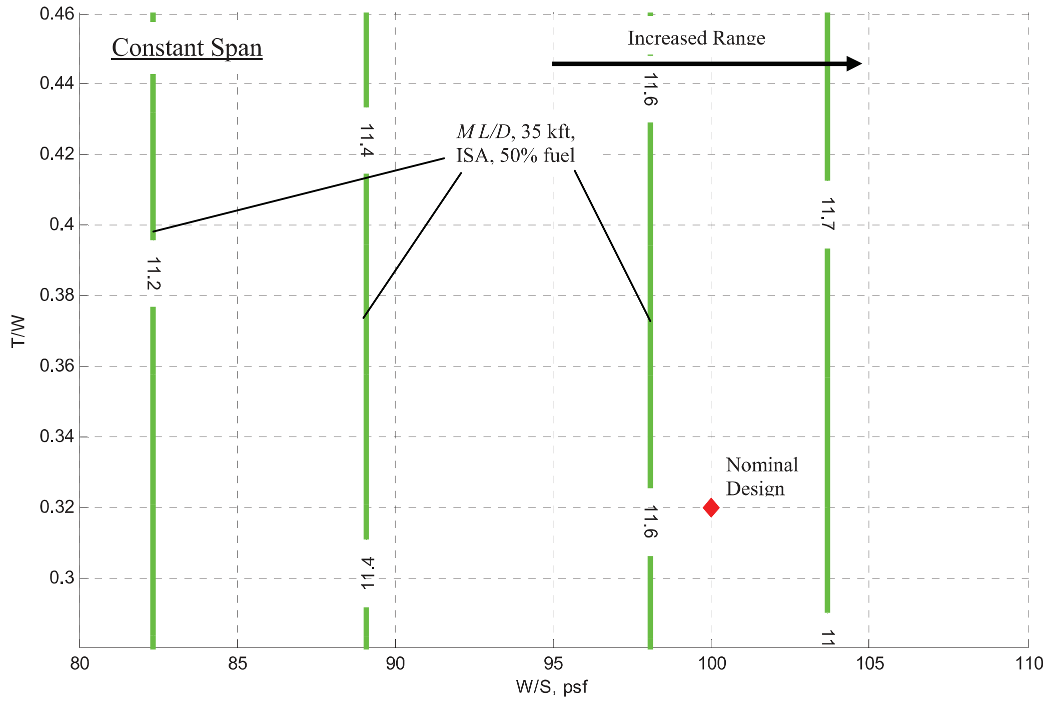

Figure 5. Constant span wing loading and thrust ratio carpet. Range parameter M·L/D curves.

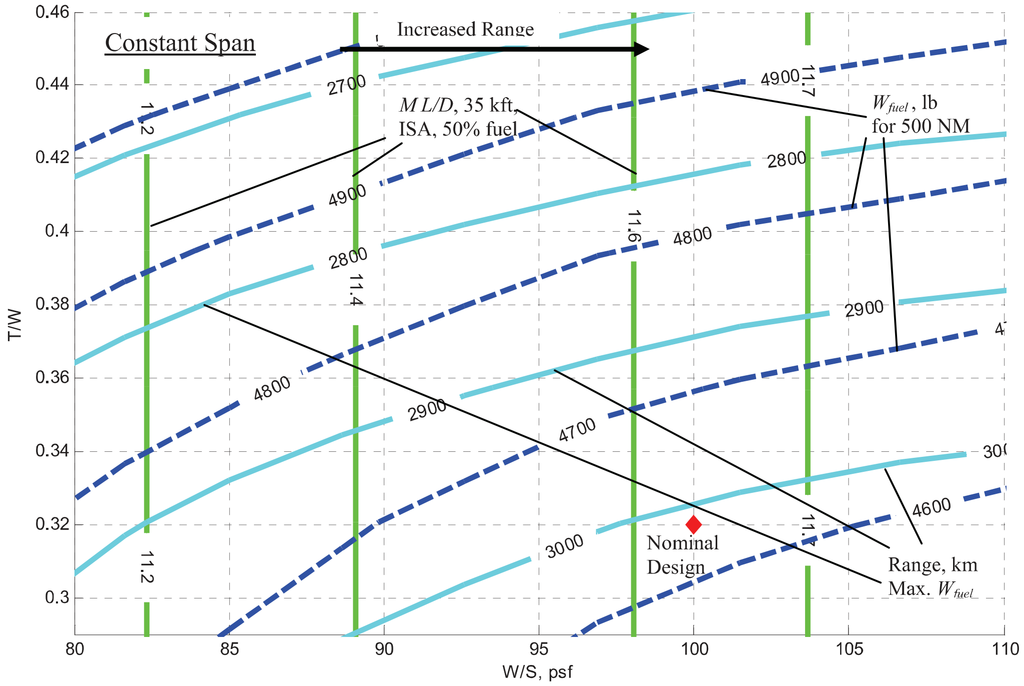

Figure 6. Constant span wing loading and thrust ratio carpet. Range parameter *M·L/D*, 100% fuel range, and 500 NM fuel curves.

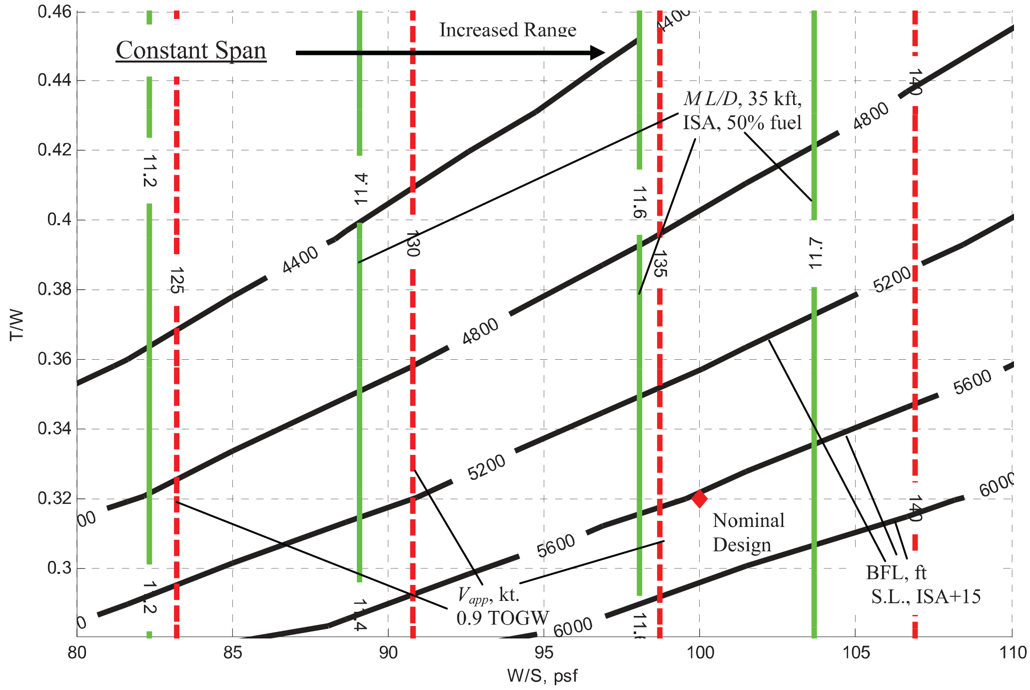

Figure 7. Constant span wing loading and thrust ratio carpet. BFL, V_{app}, and range parameter *M·L/D* curves.

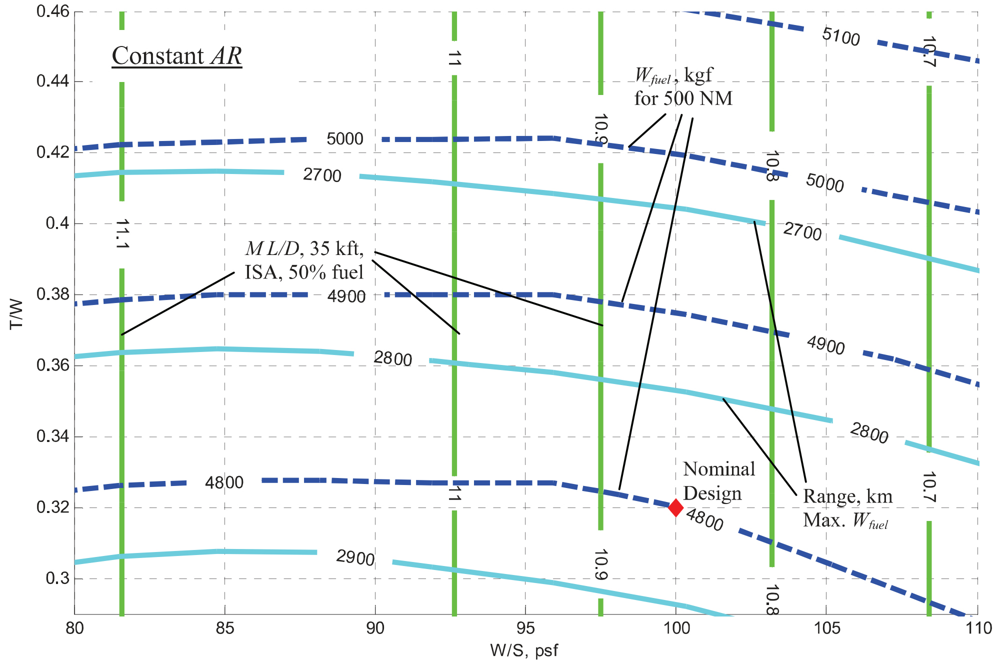

Figure 8. Constant AR wing loading and thrust ratio carpet. Range parameter *M·L/D*, 100% fuel range, and 500 NM fuel curves.

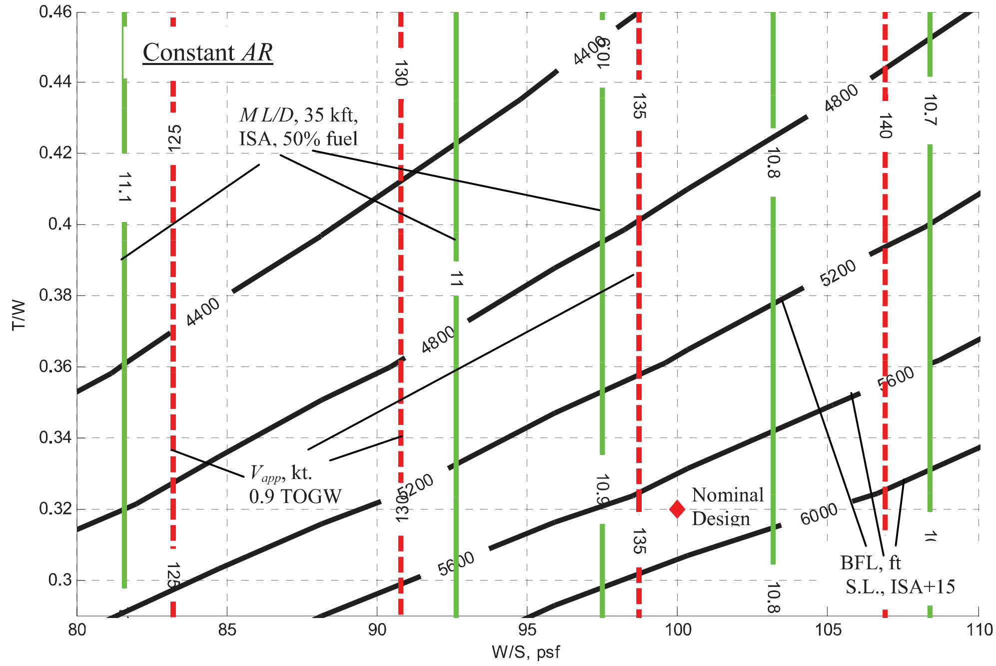

Figure 9. Constant AR wing loading and thrust ratio carpet. BFL, V_{app}, and range parameter *M·L/D* curves.

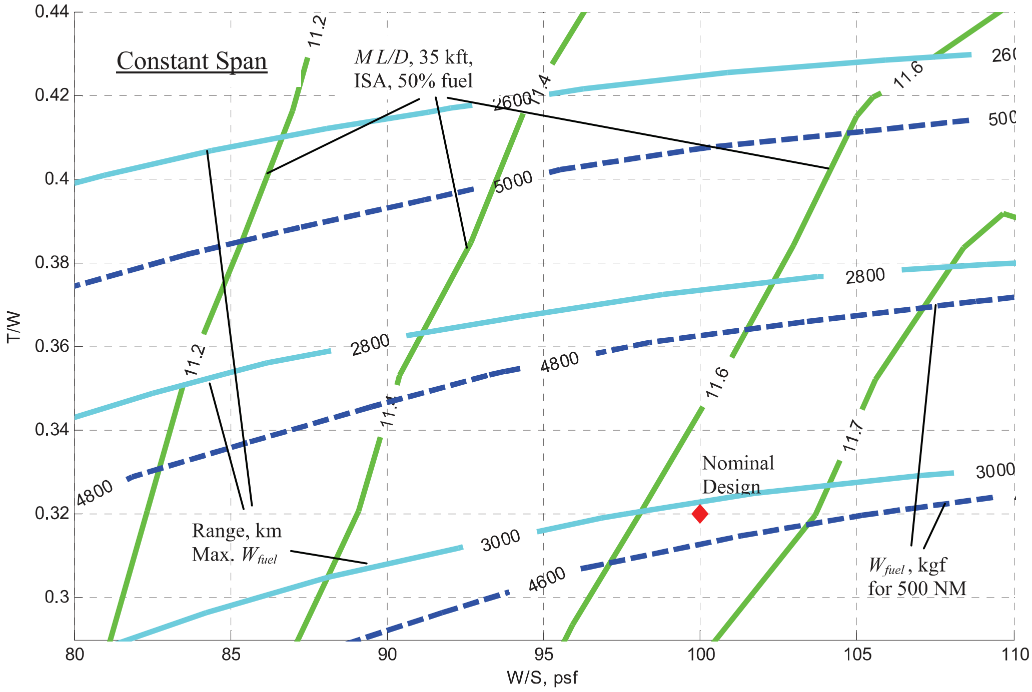

Figure 10. Constant span wing loading and thrust ratio carpet. Range parameter *M·L/D*, 100% fuel range, and 500 NM fuel curves. Thrust factor is cross-influenced with the weight model.